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# FOREIGN TECHNOLOGY DIVISION

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## AERODYNAMIC AND THERMODYNAMIC CHARACTERISTICS OF SUPERSONIC RAMJET ENGINE WITH MACH NUMBER BETWEEN 3 AND 7
## PART II - CALCULATION AND ANALYSIS OF THERMODYNAMIC PROPERTIES

By

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FOREIGN TECHNOLOGY DIVISION

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Aerodynamic and Thermodynamic Characteristics of Supersonic Ramjet Engine with Mach Number Between 3 and 7.

Part II - Calculation and Analysis of Thermodynamic Properties

(Revised Manuscript, September, 1964)

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## I. Foreward

In 1962 we made a preliminary analysis [2] on the generalized aerodynamic and thermal properties of hydrogen-fueled high Mach number ramjet engines based on the thermodynamic properties and computation method given in Reference [1]. Since more detailed characterization [3] became available on the thermal properties of hydrogen-air combustion products, we have performed computation checks and extended our calculation to the effects of non-equilibrium flow [4]. In the meantime, a master curve method was developed in the course of improving our calculation scheme and the characteristics of equilibrium and nonequilibrium expansion of hydrocarbon fuel has been calculated. (See Part Ia in appendix).

This report gives a composite account of the calculation results mentioned above and discusses the following major problems:

(1) Investigate the effects on the aerodynamic and thermodynamic characteristics of the engine by various working parameters and design parameters such as the Mach number, the recovery coefficient of inlet pressure, the type of fuel - hydrogen or hydrocarbon, the degree of expansion, the nonequilibrium expansion of high temperature combustion gas, and so on.

(2) The effects of pressure increase under the wing.

(3) Investigate the operating parameters, aerodynamic and thermal properties, and adjustment range of various major components of the engine under the requirements of maximum thermal efficiency and thrust coefficient while keeping the design relatively simple (e.g. unadjustable nozzle throat).

Based on the analysis and calculation of the above aspects, this paper presents an evaluation of the conclusions found in the literature dealing with the performance and application of ramjet engines. We also present some preliminary information on key questions in research of high Mach number ramjet engines and its aerodynamic and thermodynamic properties.

$\underline{A}$ Cross-sectional area of passage

$a$ Sound velocity, meters/sec

$C_F$ Thrust coefficient ($\equiv F_t/\rho_1 W_1 A_e/2g_c$

$C_W$ Velocity coefficient

$f$ Fuel to air weight ratio

$F_t$ Thrust, Kg

$g_c$ 9.81 Kgm/Nsec$^2$

$H$ Flight altitude

$H_f$ Same as $J(-i_{RP})$, heat release of unit mass of fuel

$I_a$ Specific impulse of air, N/(Kg/sec)

$I_f$ Specific impulse of fuel, N/(Kg/sec)

$i$ enthalpy Kcal/Kg

$J$ Mechanical equivalent of heat, 426.9 $N^m$/Kcal

$M$ Mach number

$P$ Pressure, N/cm$^2$

$q$ $\rho W^2/2g_c$

$s$ Entropy Kcal/Kg°K

$T$ Absolute temperature, °K

$W$ Relative flow velocity, m/sec

$\beta$ Fuel coefficient or fuel equivalent ratio

$\delta$ Deflection angle of air flow around the shock wave at the front of the wing, with engine mounted under the wing.

$\gamma$ Specific heat ratio

$\sigma$ back pressure recovery coefficient

$\eta_c$ combustion efficiency

$\eta_{K.E.}$ Kinetic energy efficiency of inlet

$\eta_0$ Total efficiency of engine

<u>Subscripts</u> -------, various cross sections of the engine. See (a)-(f) in Figure 27 of Ref. (1).

0 static parameters

* critical parameters

c combustion chamber of complete expansion

d inlet

e equilibrium

f expansion under fixed constituents and equilibrium vibration

ff expansion under fixed constituents and vibration

i incomplete expansion

s isentropic process

## II. Original Data Used in Characteristics Calculations

### (1). Flight path

The great majority of calculations (throughout Ref.[2] and [4] and also in part of Ref. [5]) assumed a flight path where H is equal to 24, 31 and 26 Km for M of 3, 5 and 7 respectively. Judging from the data compared in Figure 5 of Ref [1], the flight starts from some H-M point already reached and approaches $q_1 = 0$ (illegible) and the cruise flight path suggested by [11] in Reference [1]. Part of the calculations in Reference [5] assumes a flight path of $p_3 0$ between 1 and (illegible) atmospheres (see Figure 2 in Ia).

### (2). Inlet state 1 and 1°θ

Table 1 gives the atmospheric parameters and the static state for Mach numbers 3, 5 and 7 on the first type of flights described above. The atmospheric parameters were based on old data which are somewhat different from those in I. The static state is based on curves in Reference [6] and is less accurate compared to that obtained from the thermodynamic table in I.

Table 1. Inlet State

| $M_1$ | 3 | 5 | 7 |
|---|---|---|---|
| H [公里](1) | 24 | 31 | 36 |
| $p_1$ [公斤/厘米²](2) | 0.0309 | 0.01093 | 0.00333 |
| 1 [千克/米²](3) | 0.0482 | 0.01605 | 0.0074 |
| $T_1$ [°K] | 219 | 233 | 246 |
| $p_{1o}$ [公斤/厘米²](4) | 1 | 5.5 | 29 |
| $T_{1g}$ [°K] | 600 | 1300 | 2300 |
| $a_1$ [米/秒](5) | 296 | 306.4 | 315.3 |
| $i_1$ [千卡/千克](6) | 146.5 | 334 | 640 |

Key: 1 - Km; 2 - N/cm$^2$; 3 - Kg/m$^2$; 4 - N/cm$^2$; 5- m/sec; 6 - Kcal/Kg.

(3) Recovery coefficient of inlet back pressure

The $\sigma$- $M_1$ curves found in the literature show considerable differences (see Figure 11 in I). Based on the experimental data reported in Reference [7] and [8], we compiled the curve in the figure( $\sigma$actual) and take it to be the maximum value of $\sigma_d$ that can be obtained to date. In addition, we have also chosen the upper and lower limits of $\sigma_d$: the upper limit curve is from the isentropic, variable $\gamma$ expansion pressure inlet of Reference [9] and the lower limit curve corresponds to the double cone inlet of Reference [10]. (However, for $M_1 < 4$, the $\sigma_d$ value given in Reference [10] seems too high, we therefore use the $\eta_{K.E.} = 0$ multiple wave inlet data of Reference [11]).

In order to reduce the amount of computation, we have only computed( $\sigma$actual) for the case of pressure increase under the wing. For this case ( $\sigma$actual) is obtained from the curves described above and the Mach number behind the lower shock wave. Using the results of Reference [12] and choosing the shock wave deflection angle of $\delta = 10$, various parameters for the flow passing the shock wave at different $M_1$ values can all be obtained from the curves of Reference [6]. (The effects of real gas properties of air such as variable specific heat have been taken into account).

(4) <u>Back pressure recovery coefficient of the combustion chamber</u> $\sigma_c$

$\sigma_c$ is estimated from the Mach number $M_2 = 0 \cdot 15$ at the combustion chamber inlet for a constant diameter chamber with temperature increase and friction loss taken into consideration. The values of $\sigma_c$ are listed in Table 2.

Table 2.

$\sigma_c$

| $M_1$ \ $\beta$ | 0 • 5 | 1 • 0 | 2 • 0 |
|---|---|---|---|
| 3 | 0 • 95 | 0 • 94 | 0 • 945 |
| 5 | 0 • 965 | 0 • 96 | 0 • 965 |
| 7 | 0 • 98 | 0 • 97 | 0 • 98 |

(5) <u>Combustion Chamber Outlet Back Pressure</u> $p_3^*$.

Based on the choices of data and the procedure just described, the $p_3^*/p_{01}(=\sigma)$ values obtained are given in Tables 3A and 3B.

Table 3a. σ values without pressure increase under the wing

| $M_1$ | 3 | | | 5 | | | 7 | | |
|---|---|---|---|---|---|---|---|---|---|
| $\beta$ | 0.5 | 1 | 2 | 0.5 | 1 | 2 | 0.5 | 1 | 2 |
| $\sigma_{d,actual}$ | 0.69 | | | 0.3 | | | 0.14 | | |
| $\sigma_c$ | 0.95 | 0.94 | .945 | .965 | .96 | .965 | .968 | .97 | .98 |
| $\sigma_{d,high}$ | 0.9 | | | 0.48 | | | 0.22 | | |
| $\sigma_{d,low}$ | | | | 0.2 | | | 0.07 | | |
| $\sigma_{high}$ | .855 | .845 | .85 | .463 | .461 | .463 | .216 | .214 | .216 |
| $\sigma_{actual}$ | .655 | .648 | .652 | .289 | .288 | .289 | .137 | .136 | .137 |
| $\sigma_{low}$ | | | | 0.193 | | | 0.069 | | |

Table 3b. σ values with pressure increase under the wing

| $M_1$ | = 3 | | | = 5 | | | = 7 | | |
|---|---|---|---|---|---|---|---|---|---|
| $\beta$ | 0.5 | 1.0 | 2.0 | 0.5 | 1.0 | 2.0 | 0.5 | 1.0 | 2.0 |
| $\sigma_{shock\ wave}$ | 0.965 | | | 0.91 | | | 0.77 | | |
| $\sigma_{d,actual}$ | 0.835 | | | 0.46 | | | 0.28 | | |
| $\sigma_c$ | .95 | .94 | .945 | .965 | .96 | .965 | .98 | .97 | .98 |
| $\sigma_{shock\ wave} \cdot \sigma_d$ | 0.806 | | | 0.418 | | | 0.215 | | |
| $\sigma$ | .766 | .758 | .762 | .404 | .402 | .404 | .211 | .209 | .211 |

(6) Fuel

Most of the calculations were made for $H_2$ fuel because of the following reasons (1) this fuel provides a greater specific thrust, (2) it is favorable for regenerative cooling under high $M_1$ conditions, (3) the flow of the hot combustion products in the tail ejector is close to being an equilibrium expansion process. (IV) Reference [11] has made rather complete analysis for kerosene fuel where as similar information for $H_2$ fuel has been lacking, (V). In Reference [13] - [16], calculations made on $H_2$ fuel ramjet engines of different characteristics have led to quite different results regarding launch payload capacity and economy. In order to evaluate these differing conclusions, one must judge upon the feasibility of the various characteristics of the $H_2$ fuel ramjet engine assumed in these references. In this calculation, the heat release of $H_2$ is taken to be $H_f = J(-i_{RP}) = 12.1 \times 10^6$; and in the calculation for $(CH_2)_L$ fuel, $H_f$ is taken to be $4.419 \times 10^6$ N-m/Kg.

(7) <u>Calculations of thrust and thermodynamics properties</u>

The following equation is still widely used in current references for the calculation for supersonic ramjet engines.

$$F_t = \frac{(1+f)G_1 W_e}{g_c} - \frac{G_1 W_1}{g_c} + (p_e - p_1) \cdot A_e \qquad (1)$$

It has been pointed out in section II.5 of I that the above equation represents net thrust only when the flow outside the airplane body is isentropic. This condition is approximately satisfied only for subsonic and, for supersonic flight, there is invariably a shock wave system outside the plane body. Therefore, the effective thrust under supersonic conditions generally can no longer be found from Equation (1) but instead is dependent upon the actual shape of the airplane body. For the sake of comparison with data reported in the literature and avoiding the difficulties associated with resistance computation for a particular body design, we will consistently resort to the following calculations:

Without using pressure increase under the wing

$$F_{t,i} = \frac{(1+f)G_1 W_e}{g_c} - \frac{G_1 W_1}{g_c} + (p_e A_e - p_1 A_1) \tag{2}$$

With pressure increase under the wing

$$F_{t,i} = \frac{(1+f)G_1 W_e}{g_c} + p_e A_e - \left(\frac{G_1 W_1}{g_c} + p_1 A_1\right) \cos\delta \tag{3}$$

(Derivatives of Eqs. (2) and (3) can be found in Section II.4 of I).

In Eq. (3), parameters for cross section 1 can be computed from shock waves under the wing and it is also assumed that the direction of the exhaust and the resulting <u>internal thrust</u> are both parallel to the direction of the flight.

Equations for other relevant thermodynamic properties are:

$$I_{a,i} = F_{t,i}/G_1 = \frac{1}{g_c}\left[(1+f)\, W_e - W_1\right] \tag{4}$$

$$I_{f,i} = \frac{I_{a,i}}{f/\eta_c} = \frac{W_1^2\left[(1+f)\left(\frac{W_e}{W_1}\right)^2 - 1\right]}{2g_c f H_f/\eta_c} \tag{5}$$

$$\eta_{o,i} = \frac{I_{f,i}\, W_1}{H_f} \tag{6}$$

$$C_{F,i} = \frac{F_{t,i}}{\frac{1}{2g_c}\rho_1 W_1 A_e} = 2\left[(1+f)\frac{W_e}{W_1} - 1\right]\frac{A_1}{A_e} \tag{7}$$

$$\frac{F_{t,i}}{p_1 A_1} = 1.4\, M_1^2\left[(1+f)\,\frac{W_e}{W_1} - 1\right] \tag{8}$$

$\eta_c$ is taken to be 1 in all calculations here. When comparisons are to be made with other authors, the results contained in this paper need to be converted assuming $\eta_c = 0.95$ and calculating the thrust according to Eq. (1). The total efficiency, as defined in Reference [11], is given by

$$\eta_o' = \frac{I_f \; W_1}{H_f + \frac{W_1^2}{2g_c}}$$

therefore, in comparison with the results of Reference [11], we convert the values in [11] as follows:

$$\eta_o = \left(1 + \frac{W_1^2}{2g_c H_f}\right) \eta_o' = C_\eta \, \eta_o'$$

where $2\, g_c H_f = 2x32.2x778x18630 = 934 \times 10^6$.

We obtained the conversion coefficient $C_\eta$ for flight paths with dynamic pressure $q_1 = \frac{\rho_1 W_1^2}{2g_c} = 350 \frac{lb}{ft^2}$ as given in Reference [11]. The values of $C_\eta$ are tabulated below.

Table 4. $C_\eta$ values for $q_1 = 350\ lb/ft^2$

| $K_1$ | 3 | 4 | 5 | 6 | 7 | 8 | 9 | 10 |
|---|---|---|---|---|---|---|---|---|
| $C_\eta$ | 1.009 | 1.017 | 1.028 | 1.041 | 1.057 | 1.076 | 1.090 | 1.126 |

(8) Complete expansion and incomplete expansion

We use the following definitions for complete expansion and incomplete expansion.

Complete expansion - when the gas pressure $p_e$ over the cross section of the tail ejector pipe outlet is equal to the ambient atmosphere pressure $p_1$. Actually, in supersonic flights, the back pressure behind the tail ejector pipe is not equal to $p_1$ and this is especially true when there is a pressure increase under the wing. For this reason, the definition given here for complete expansion should be regarded as a comparative standard.

Incomplete expansion - consider only one type of tail ejector, the ratio of the outlet cross sectional area and the gas collecting area $Ae/A_1 = 1.5$. The ratio is appropriate for the exterior design and resistance where the under wing pressure increase is not used. It is also convenient for comparing the results here and those of Reference [11].

We calculated the properties for complete expansion because generally one is likely to achieve complete expansion using the body and wing area facing the wind while not necessarily increasing the external resistance.

(9) Equilibrium and nonequilibrium expansion processes

We have made calculations for hydrogen fuel and hydrocarbon fuel under complete expansion for three expansion processes: expansion process where both the components and vibration are kept under equilibrium, expansion where components are frozen (from 3°° and up) and vibrations are in equilibrium, and finally expansion process where both are frozen. (Methods of computation are given in Section II.4 in I).

Over the upstream of the ejector throat, flow is assumed to be isentropic. $C_W = 0.97$ in calculating the outlet nozzle velocity We.

(10) Calculation Results

We have made computations for 20 design points for the configuration without under wing pressurization. Thermodynamic properties of the engine and major aerodynamic and thermal parameters for various cross sectional areas are summarized in the attached tables.(Table 1 and Table 2). We have plotted these results against $\beta$ to observe the $\beta$ values when $\eta_{0,i}$ and $C_{F,i}$ reach their respective maximum values for different $M_1$, see Figures 1, 3 and 4. The variation from maximum efficiency to maximum thrust coefficient is obtained for different $M_1$ values by plotting the aerodynamic thermal parameter and thermodynamic properties just obtained versus $M_1$, see Figures 2 and 5.

III. Effects of $M_1$ and $\beta$ on aerodynamic thermal parameter and thermodynamic properties

To observe the $M_1$ and $\beta$ effects on the aerodynamic thermal parameter and on the thermodynamic characteristics of the engine, we first plot the data [5] (in attached Table 1) for the case of no pressurization under the wing versus $\beta$. See Figure 1a-1e. It can be seen from the figure that for small $\beta$, the gas temperature $T_s^0$ increases rapidly with increasing $\beta$. (The increase is less rapid for large $\beta$). The rise of $T_3^{\circ}$ slows down as $\beta$ approaches 1 because of the boundary at high temperatures. $T_3^0$ reaches its maximum in the vicinity of $\beta \approx 1.1$, while $T_3^0$ follows the increasing $\beta$ all along. Under these conditions, the increase in We is rather small (see Figure 1b) after $\beta > 1.28$ even if combustion gas can achieve complete equilibrium expansion in the exhaust nozzle. In certain cases, We can even decrease.

For the same $\beta$, $T_3^{\circ}$ and We increase with $M_1$, since $T_1^0$ increases with $M_1$. In the meantime, both (We - $W_1$) and We/$W_1$ are decreasing with $M_1$ as shown in Figure 1c. As a result, $I_{a_1 i}$ and $I_{f_1 i}$ decreases with $M_1$, as shown in Figures 1d and 1e. Figure 1e also shows that $I_{f_1 i}$ reaches its maximum for larger $\beta$ as $M_1$ increases, however the maximum $\beta$ value ( ~0.6) is still considerably less than 1 even for $M_1 = 7$.

Since $\eta_{o,i}$ is proportional to the product of $I_{f_1 i}$ and $W_1$, $\eta_{o,i}$ increases with increasing $M_1$ and for a given $M_1$, the $\beta$ value for maximum $\eta_{o,i}$ is the same as for $I_{f_1 i}$. Calculations for hydrogen fuel under equilibrium flow and complete expansion conditions lead to the following approximate $\beta$ values at maximum $\eta_{o,i,e}$ :

| | | | |
|---|---|---|---|
| $M_1$ : | 3 | 5 | 7 |
| $\beta_{max.\eta_{o,i,e}}$ : | $\gtrsim 0.1$ | 0.3 | 0.6 |

The value of $Ae/A_1$ depicted in Figure 1g is determined mainly by $W_1/W_e$ (Figure 1c) and $\frac{\rho_1}{\rho_e}\left[\approx \frac{p_1}{p_0}\frac{T_e}{T_1}\right]$. $M_1$ has only a small effect on $Ae/A_1$ and the variation with is similar to that of We; as a result, the changes in $C_{F,i}$ are similar to those of $We/W_1$. For $M_1 = 5$ and $M_1 = 7$ the variations are extremely small for $\beta$ greater than 1.5.

Figure 1i shows the variation of yet another thrust coefficient $F_t(p_1A_1)$ whose value is determined mainly by $M_1{}^2$ and $We/W_1$. Therefore, the variation of $F_t/p_1A_1$ with $\beta$ is similar to that of $We/W_1$ for the same $M_1$ and increase with $M_1$ for the same $\beta$.

The variation [2] of $A_1/A_2$ as a function of $M_1$ is shown in Figure 1j. Because the rate of increase of the density ratio $(\rho_2/\rho_1)$ is much faster than that of velocity ratio $(W_1/W_2)$, $A_1/A_2$ increases rapidly as $M_1$ gets larger. It seems that, for an engine with constant $A_2$, the values of $A_1$, needs to be increased 7 times as $M_1$ goes from 3 to 7. It can also be seen from Figures 1k and 1l that $\beta$ does not large effects on $A_{3*}/A_2$ and $Ae/A_2$, however, Ae again needs to be increased by a factor of 7 as M increases from 3 to 7.

## IV. Aerodynamic and thermodynamic properties at maximum $\eta_{o,i,e}$ and maximum $C_{F,i,e}$ for different M numbers

From the three curves for $M_1$ equals 3, 5 and 7 in Figure 1, we have chosen 0.1, 0.3, 0.6 and 2.5 as the $\beta$ values for maximum $\eta_{o,i,e}$ and maximum $C_{F,i,e}$ and indicated these in Figure 2. Figure 2A indicates that, as $M_1$ increases, the maximization of both $\eta_{o,i,e}$ and $C_{F,i,e}$ requires $T_3 0$ to increase, although the former has a much smaller effect. When $M_1$ is slightly greater than 7, $T_3 0$ for both are the same even though the corresponding $\beta$ values are far apart ($\sim 0.6$ and $\sim 2$). This is because, at high temperatures, $T_3 o$ decreases as $\beta$ increases (see Figure 1A). If the flow can be kept in equilibrium, the $i_3 0$ value at high $T_3 0$ can still increase the We value. Therefore, when $M_1 = 7$, the difference in We for maximum $\eta_{o,i,e}$ and for maximum $C_{F,i,e}$ is greater than that of $T_30$. From Figure 2C, we can see that, as $M_1$ increases from 3 to 7, the value of $We/W_1$ for maximum $\eta_{o,i,e}$ is almost a constant 1.2 whereas the $We/W_1$ for maximum $C_{F,i,e}$ has decreased from 2.5 to 1.4. This variation of $We/W_1$ basically determines the variations of maximum $I_{f,i,e}$ and maximum $\eta_{o,i,e}$ with $M_1$. From more than 6000, the maximum $I_{f,i,e}$ decreases by a factor of 2 (Figure 2f) while maximum $\eta_{o,i}$ continues to climb with $M_1$ and reaches its peak value of 0.626 near $M_1 = 7$ (Figure 2g).

Since the $Ae/A_1$ for maximum $C_{F,i,e}$ increases very slightly with $M_1$, the maximum $C_{F,i,e}$, like $We/W_1$, decreases rapidly with $M_1$ (from 0.88 to 0.24). For high $M_1$ values, one effective way to increase $C_F$ is to use incomplete expansion and thus a reduced Ae.

## V. Inlet Stopping-Pressure Recovery Coefficient and Completeness of Expansion - their Effects on the Aerodynamic and Thermodynamic Properties

The influence of high $\sigma_d$ value and low $\sigma_d$ value on the thermodynamic properties can be realized in Figure 3 (computation of Reference [2]) and Figure 6. For $M_1$ greater than 5, the increasing $\sigma_d$ has a gradually diminishing effect on $\eta_{o,i,e}$ and $I_{f,i,e}$. There is only about 5% increase in $\eta_{o,i,e}$ or $I_{f,i,e}$ as $\sigma$ increases from $\sigma$ actual to $\sigma$ high. (The effect is greater under nonequilibrium conditions at high temperatures.)

Figures 3 and 5 show the effects of increasing $C_F$, decreasing nozzle outlet cross sectional area Ae on the completeness of expansion. Fixing the Ae value at 1.5A, for all $M_1$ values leads to the following results (1) increase of Ae with $M_1$ is very small (Figure 3d), (2) We decreases, the amount of decrease is smaller for high $M_1$ than for low $M_1$ (Figure 3A), (3) $C_{F,i,e}$ increases, the amount of increase is smaller at higher $M_1$, see Figure 3c, (4) $\eta_{o,i,e}$ decreases, the amount of decrease is greater for higher $M_1$ except when $\beta > 1.5$. We noticed that, when $M_1 = 7$ and $\beta = 0.5$, setting $A_e$ equal to 1.5A, will decrease $\eta_{o,i,e}$ by 28%.

## VI. Effects of Fuel and Nonequilibrium Expansion

Although the discussion so far has been primarily based on the calculation results for hydrogen fuel,[2,4] the data for hydrocarbon fuel under the same flight conditions[5] have also been included in Figure 1. The following observations can be made from the graph.

(1) For the same $T_10$ and $\beta$, the $T_30$ values for the two types of fuel are very close to being equal even when the fuel amounts differ by more than a factor of 2 .

(2) For the same $T_10$ and $\beta$, $(i_30 - i_e)$ for hydrogen fuel is greater than that for hydrocarbon fuel and the difference gets larger at higher temperatures. Thus, the equilibrium expansion ejection speed of hydrogen fuel is greater compared to hydrocarbon fuel, and the difference in ejection speed increases as $M_1$ or $T_10$ increases (Figures 1B and 1C).

(3) The low $I_a$ value of hydrocarbon is due to its lower We, see Figure 1d; however, because hydrocarbon has a lower $fH_f$ than hydrogen, the result is that the overall efficiencies $\eta_{o,i,e}$ for both fuel are quite close (Figure 1f).

(4) Because of the lower $I_a$ and higher (more than twice) f of hydrocarbon fuel, its $\bar{I}_{f,i,e}$ is lower than that of hydrogen by more than a factor of 2 for the same $M_1$ and $\beta$ . (Figure 1e).

(5) For the same $T_30$ and $\beta$, the molecular weight of hydrocarbon fuel gas is higher than the molecular weight of hydrogen

by approximately 20% (Figure 7). Thus, for the same $T_e$ and $P_e$ the density of the former is 20% higher than the latter. The ejector cross-sectional areas of the former are 20% smaller than those of the latter for the same flow rate and ejection speed. Judging from Figure 1g, the ejector cross section for hydrocarbon fuel is approximately 10% smaller than its counterpart for hydrogen for the same $M_1$ and $\beta$.

Perhaps another major difference between the two types of fuels can be seen from the effects of nonequilibrium flow. Fig. 8B indicates the effects of nonequilibrium expansion on the characteristics of the two fuels upon approaching maximum $\eta_{o,i,e}$ under various $M_1$ conditions. For high $M_1$ and high $\beta$, the percentage decrease in $W_e$ of hydrocarbon due to nonequilibrium flow is not much larger than for hydrogen fuel, nevertheless, this decrease is weighted heavier for $W_e - W_1$, and, as a result, the decreases in $I_{ai}$, $I_{f_1 i}$, $\eta_{o,i}$ are larger for hydrocarbons. See Figure 8d-8f.

## VII. Influence of altitude

As an example in $I_a$, we obtained the aerodynamic and thermodynamic properties of the engine when $p_{3o}$ is kept between 1 and 10 atm., See Figures 8-10 in Ia. The equilibrium and frozen-constituent $\eta_o$ and $I_f$ versus $M_1$, under maximum $\eta_{o,e}$ and maximum $I_{f,e}$, as shown in Figure 9, can be obtained from the data in Figures 8-10 using the envelope curve of different $\beta$'s or plotting $\eta_o$ and $I_f$ against $\beta$. Figure 9 clearly shows the advantages of lower flight altitudes at high $M_1$ -- not only $\eta_{o,e}$, and $I_{f,e}$ are increased, the effects of nonequilibrium flow are also substantially reduced. It seems that reducing the flight altitude is an effective means of minimizing the nonequilibrium effects at high $M_1$.

## VIII. Comparisons of thermodynamic properties of partially adjustable and totally adjustable engines and their adjustment criteria.

Since all our calculations are made under the assumption that components achieve their optimum performance for a given $M_1$, these calculations should thus be viewed as corresponding to engine performance where the geometric configuration is totally adjustable to meet optimization criteria when there is a variation in $M_1$. In this section, we will make some analysis and comparisons of the component adjustment criteria and their corresponding thermodynamic performances for the following three situations: (a) requiring the maximum efficiency $\eta_{o,i,e}$ for all flight Mach numbers $M_1$, (b) requiring the maximum thrust coefficient $C_{F,i,e}$ for all $M_1$, and (c) allowing a constant cross-sectional area at the ejector throat.

(1) Maintaining maximum efficiency

First, the maximum efficiency $\beta$ - $M_1$ curve (the $\beta_\eta$ curve in Figure 5a) is obtained from the $\eta_{o,i,e}$ curves for various $M_1$ under complete expansion in Figure 3. The variation pattern of $\eta_{o,i,e}$ , $C_{F,i,e}$ , $A_3{}^*/A_2$, $T_3 0$, $F_{G*,i}$ and $A_1/A_2$ for maximum efficiency can be found from other curves in Figure 3. Results are shown as thick solid curves in Figure 5.

(2) Maintaining maximum thrust coefficient

The $\beta$ - $M_1$ curve corresponding to maximum $C_{F,i,e}$ is obtained from the $C_{F,i,e}-\beta$ curves for different $M_1$ in Figure 3 where incomplete expansion with $A_e/A_1 = 1.5$ is assumed. The result is the $\beta_{C_F}$ line in Figure 5c. The variations of other parameters under the maximum $C_F$ condition can then be obtained from other curves in Figure 3. See thick dashed line in Figure 5.

(3) Keeping $A_3{}^*$ constant

The thick dash-and-dot curve in Figure 5 shows the variation of parameters by using Figure 3 and setting $A_3{}^* = 0.34\ A_2$.

In view of the variation pattern of various parameters under the three adjustment conditions listed above, the constant ejector throat scheme is only slightly behind the maximum efficiency scheme in its effectiveness, thrust performance, $T_3{}^0$ and $A_e$ while maintaining its advantage of not having to adjust the throat. This comes about because the maximum efficiency scheme does not require any large changes in $A_{3*}$. (see Figure 5f). As compared to scheme (2), scheme (3) has high efficiency (upper 70%), is easy to adjust and its combustion temperature $T_3{}^0$ is also low. A low $T_3{}^0$ implies a relatively small boundary effect in real flow and less problems with heat transfer and material structure. Disadvantages are a lower $C_{F,i}$ and relatively large adjustments in outlet crosssection e.

## IX. Thermodynamic performance and adjustment criteria of engines with under wing pressurization.

Table 2 (attached) and Figure 4 are compilations of calculation results when under wing pressurization is used. Based on these data, the engine performance under the three adjustment conditions discussed above can be arrived at in a similar fashion to the case without under wing pressurization. These results are included in Figure 5 as five lines [Note - the computation of $C_{F,i}$ with pressurization is based on the complete expansion assumption].

(1) Effects of underwing pressurization

The advantages, as can be seen From Figure 5, are

(a) The increase in $\eta_{o,i,e}$ is large. The amount of increase and the percentage increase are both increasing with $M_1$ (Figure 5b). For example, for $M_1 = 7$, the efficiency gained through pressurization is equal to

$$\Delta \eta_{max} / \eta_{o,max.} = (0.79 - 0.615) \div 0.615 = 0.285.$$

The increase in efficiency is probably caused by two factors. First, the total $\sigma$-value is increased because of the additional shock wave from the pressurization. Secondly, the resistance produced by the first wave is included in that of the wing. Since there are always shock waves produced by the wing, the net effect is equivalent to circumventing the resistance of the first wave.

(b) Maximum $C_{F,i,e}$ (complete expansion) has a slight increase, see Figure 5c.

(c) There is little variation in $A_1/A_2$. This is more evident at higher $M_1$ (see Figure 5e). As a result, the adjustment of inlet is relatively straightforward at high M numbers.

(d) If one uses the $\eta_{o,i} = \eta_{o,max}$ scheme, the variation in exhaust cross section $A_e$ is very small, see Figure 5e.

On the other hand, pressurization also brings about some disadvantages:

(a) If the $\eta_{o,i} = \eta_{o,i\,max}$ scheme is used, the variation required in the throat cross section is larger, see Figure 5f.

(b) If one uses the $A_3^* =$ constant scheme, no problems are encountered for $M_1 < 6$ where adjustments are simple and $\eta_{o,i}$ is high. For $M_1 > 6$, however, the decrease of $\eta_{o,i,e}$ is rapid. The $A_3^* =$ constant scheme is therefore unsuitable for $M_1 > 6$. Without pressurization, this scheme is acceptable because of its satisfactory $\eta_{o,i,e}$ at high $M_1$.

An overall view indicates that pressurization under the wing does not change the performance parameters in any qualitative way, its effect is mainly quantitative and usually leads to improvements. Generally speaking, the advantages are more prominent at higher $M_1$. Naturally, in order to realize the advantages in practice, considerations must be given to the design configuration of the engine, wings and body. For instance, since the angle is smaller at higher $M_1$, the engine inlet must be placed farther back from the front edges of the wings. Detailed analysis on this problem can be found in Reference [12].

(2) Comparison of the three adjustment schemes

In the $A_{3*}$ = constant scheme, the adjustments required are small and thermodynamic properties are good. $\eta_{o,i,e}$ decreases rapidly for $M_1$ greater than 6.

In the maximum $C_{F,i,e}$ scheme, $A_3$* is required to vary over a large range -- $A_3$* must be cut in half when $M_1$ increases from 5 to 7. Compared to the other two schemes, $T_30$ is much higher and this is true especially for low $M_1$.

In the maximum efficiency scheme, the required changes in $A_e$ is small and $T_30$ is also low for $M_1$ less than 7 (more prominent at lower $M_1$). These features are favorable in minimizing freezing loss and in structured strength considerations. In this scheme, the changes in $A_3$* is much less than that required in the maximum $C_{F,i,e}$ method. Another advantage is that $A_e$ is almost constant. $A_e$ can therefore be unadjustable because small variations in expansion have very little effect on the performance when total expansion is approached. From the adjustment point of view, this scheme is easy to satisfy.

## X. Comparison of subsonic combustion and supersonic combustion

The performance data of supersonic combustion engine cited in References [10] and [11] are quite close together, as shown in Figure 10. When one compares their data with our calculated results on subsonic combustion (thick line in Figure 10), one can make the following observations: For equilibrium flow and $M_1$ less than 8, a highly feasible subsonic combustion ramjet engine is superior to the supersonic combustion case. This superior tendency is estimated to hold possibly to $M_1 = 10$ if one extrapolates the curve in Figure 10. Even under the least favorable subsonic combustion conditions, as the lower six curves in Reference [19] show, supersonic combustion is still inferior for $M_1 \leqslant 6$.

Naturally, for the same altitude the subsonic combustion suffers a higher loss to the real nonequilibrium flow than the supersonic combustion does. But, as one can see from the figure, the nonequilibrium flow loss of a high $M_1$ subsonic combustion using hydrogen fuel is much less than the loss due to use of hydrocarbon fuel. The total efficiency is approximately equal to the supersonic one at $M_1$ equal to 7. At lower altitudes, $p_{3^0}$ of subsonic combustion is higher and the nonequilibrium flow loss of hydrocarbon fuel is quite small at $M_1 = 7$. It is believed that subsonic combustion ramjets can be used at lower altitudes for $M_1$ above 5.

## XI. Feasibility evaluation and comparison analysis of results in current references

We compiled the relevant data on efficiency $\eta_o$ and specific thrust $I_f$ and plotted them in Figs. 11 and 12. In these figures, the numbers assigned to each curve are the same as the reference numbers in I. Our computation results are plotted as heavy lines in the figures. We have done so to facilitate the comparison and evaluation of the ramjet engine properties found in many current references. The feasibility of using the ramjet engine as a satellite launcher and its ability can also be analyzed based on these data.

Regarding the performance of hydrogen fuel, our calculated equilibrium flow value is somewhat higher than that given by Reference [20] in I. All the results given in Reference [13] in I, in the $M_1$ = 3 to 7 range, are higher than our calculated value. As for the data in Reference [10] of I, since the same nonequilibrium flow loss has been used for all $M_1$, the performance is too low at low $M_1$; the data for $M_1$ = 7, however, seem reasonable. The performance data at low $M_1$ as reported in Reference [11] of I are even lower. Data in Reference [8] of I are too low over the entire range and the variation trend in Reference [12] is incorrect.

If $I_f$ value of hydrocarbon fuel reported in Reference [19] of I for high $\sigma$ is very close to our calculation and those reported in [10] and [11] are somewhat too low.

## XII. Concluding remarks

Based on analysis of the calculation results presented in previous sections, we can make the following preliminary deductions:

(1) When the M number increases from 3 to 7, thermodynamic properties of the ramjet engine are expected to improve further. However, when M is greater than 5.5, proper design should be made to avoid increasing losses due to nonequilibrium expansion and boundary combustion gas.

(2) When the M number reaches 7, the intake gas temperature of the combustion chamber is 2300°K and the maximum efficiency and maximum thrust require the exhaust temperature to be 3000°K.

At this temperature, the boundary effect of the combustion gas increases with decreasing pressure. At 36 km altitude, $p_{30}$ is approximately 4.4 atm. and the nonequilibrium expansion loss of hydrocarbon fuel is likely to make the efficiency at $M_1 = 7$ less than the efficiency at $M_1 = 6$. If the altitude is reduced to 29 km, then $p_{30} = 10$ atm and the nonequilibrium expansion loss is greatly reduced and the efficiency continues to increase from $M_1 = 6$ to $M_1 = 7$ and beyond. This effect is more effective than changing to hydrogen fuel. (The flight altitude can be increased to 32 km if under wing pressurization is used, see Figure 4 in Ia).

(3) It is not necessary to use supersonic combustion before $M_1$ reaches 9 or 10, hence, the research work in this area can be delayed until a few years later.

(4) The advantages of hydrogen fuel over the hydrocarbon fuel are the following: hydrogen fuel specific thrust is twice as high; although the efficiencies under equilibrium expansion are not much different, the efficiency of hydrogen fuel is much higher under actual expansion condition; and finally, hydrogen fuel is a better coolant.

(5) underwing pressurization allows the inlet channel to work at a lower and less-variable M number, more favorable for better performance.

(6) without under wing pressurization, a fixed ejector throat cross sectional area under different flight speeds has a small effect on the performance. With pressurization, the throat area must be varied according to flying speed in order to maintain the maximum efficiency over a wide range.

(7) The super high speed ramjet engine has been proposed as the second stage for a satellite launch vehicle. Its carrying capacity has been estimated in the current literature. Judging from the thermodynamic data of the ramjet engine, the esti-

mates in References [5], [6], [10] and [13] (in Part I) are all considered to be high. Those in [8] and [12] are too low and estimates in [9], [10], [19] and [20] are the most reasonable.

(8) Master curves in Ia can be used to greatly reduce the amount of aerodynamic and thermodynamic analyses similar to those presented here, and in the meantime, increase the accuracy of the analysis. These curves are even more useful for those fuels for which thermodynamic properties are not yet available.

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〔18〕即I中的〔9〕

Table 1. Aerodynamic and Thermodynamic properties of hydrogen fuel under complete expansion without under wing pressurization.

| Number | | $M_1$ | $\beta$ | $\sigma$ | $P_{30}$ | $T_{30}$ | $P_{3*}$ | $T_{3*}$ | | $T_e$ | $W_e$ | $W_e/W_1$ |
|---|---|---|---|---|---|---|---|---|---|---|---|---|
| 1 | Variation with $M_1$ and $\beta$ | 3 | .2 | .652 | .737 | 1168 | | | e | 556 | 1196 | 1.35 |
| 2 | | | .5 | | | 1821 | | | | 946 | 1567 | 1.76 |
| 3 | | | 1.0 | | | 2444 | .415 | 2264 | | 1490 | 1970 | 2.22 |
| 4 | | | 1.5 | | | 2315 | | | | 1270 | 2002 | 2.25 |
| 5 | | | 2.0 | | | 2110 | | | | 1110 | 2010 | 2.26 |
| 6 | | 5 | .2 | .289 | 1.763 | 1183 | | | | 580 | 1720 | 1.12 |
| 7 | | | .5 | | | 2206 | .971 | 2061 | | 837 | 2072 | 1.35 |
| 8 | | | 1.0 | | | 2708 | 1.002 | 2539 | | 1225 | 2498 | 1.63 |
| 9 | | | 1.5 | | | 2624 | | | | 1020 | 2520 | 1.65 |
| 10 | | | 2.0 | | | 2426 | | | | 882 | 2523 | 1.65 |
| 11 | | 7 | .5 | .137 | 4.380 | 2874 | | | | 882 | 2578 | 1.17 |
| 12 | | | 1.0 | | | 3054 | 2.476 | 2886 | | 1190 | 2966 | 1.34 |
| 13 | | | 1.5 | | | 3001 | | | | 970 | 2989 | 1.36 |
| 14 | | | 2.0 | | | 2836 | | | | 834 | 2987 | 1.35 |
| | | | | | | | | | | $T_{e,s}$ | $W_{e,s}$ | |
| 15 | Approaching maximum $\eta_c$, (Cw = 1) | 3 | .2 | .652 | .737 | 1168 | .395 | 1004 | e | 515 | 1234 | 1.39 |
| | | | | | | | | | ff | 469 | 1236 | 1.39 |
| 16 | | 5 | .5 | .289 | 1.763 | 2296 | .971 | 2061 | e | 730 | 2131 | 1.38 |
| | | | | | | | | | f | 714 | 2117 | 1.35 |
| | | | | | | | | | ff | 580 | 2033 | 1.32 |
| 17 | | 7 | .8 | .137 | 4.380 | 3018 | 2.555 | 2844 | e | 826 | 2940 | 1.33 |
| | | | | | | | | | f | 701 | 2758 | 1.21 |
| | | | | | | | | | ff | 512 | 2602 | 1.18 |
| 18 | Approaching maximum $C_F$, (Cw = 1) | 3 | 1 | .652 | .737 | 2444 | .415 | 2264 | e | 1420 | 2030 | 2.29 |
| | | | | | | | | | f | 1309 | 2003 | 2.26 |
| | | | | | | | | | ff | 1110 | 1947 | 2.19 |
| 19 | | 5 | 1 | .289 | 1.763 | 2708 | 1.002 | 2539 | e | 1130 | 2575 | 1.68 |
| | | | | | | | | | f | 938 | 2484 | 1.62 |
| | | | | | | | | | ff | 722 | 2361 | 1.54 |
| 20 | | 7 | 1 | .137 | 4.380 | 3054 | 2.476 | 2886 | e | 1010 | 3058 | 1.43 |
| | | | | | | | | | f | 725 | 2848 | 1.29 |
| | | | | | | | | | ff | 522 | 2681 | 1.22 |
| | | | | | ata | °K | ata | °K | | °K | m/s | |

| $I_a$ | $A_e/A_1$ | $I_{a,1}$ | $I_{f,1}$ | $\eta_{o,1}$ | $C_{F,1}$ | $F_{t,1}/p_1A_1$ | | | |
|---|---|---|---|---|---|---|---|---|---|
| 31.7 | 1.97 | 38.7 | 6620 | .485 | .436 | 5.3 | | | |
| 71.5 | 2.71 | 83.9 | 5740 | .422 | .685 | 10.6 | | | |
| 116 | 3.74 | 135 | 4630 | .340 | .800 | 18.8 | | | |
| 122 | 3.66 | 142 | 3230 | .237 | .853 | 19.6 | | | |
| 126 | 3.65 | 145 | 2490 | .183 | .880 | 20.0 | | | |
| 21.3 | 2.31 | 27.1 | 4650 | .587 | .151 | 6.1 | | | |
| 58.7 | 2.92 | 67.3 | 4610 | .583 | .296 | 15.1 | | | |
| 106 | 3.90 | 119 | 4080 | .516 | .392 | 26.6 | | | |
| 112 | 3.77 | 124 | 2840 | .359 | .423 | 27.7 | | | |
| 115 | 3.74 | 128 | 2180 | .276 | .438 | 28.6 | | | |
| 41.8 | 3.36 | 49.5 | 3390 | .638 | .134 | 15.1 | | | |
| 87.2 | 4.37 | 98.2 | 3360 | .614 | .200 | 29.9 | | | |
| 93.3 | 4.13 | 104 | 2360 | .430 | .223 | 31.6 | | | |
| 97.7 | 4.08 | 108 | 1842 | .336 | .236 | 32.8 | | | |
| | | | | | | | $I_a$ | $I_f$ | $\eta_o$ |
| | | | | | | | $\eta_c = .95$ | $C_w = .97$ | |
| 36.1 | 1.76 | 41.6 | 7120 | .523 | .522 | | 32.1 | 5220 | .383 |
| 36.2 | 1.60 | 40.6 | 6945 | .510 | .559 | | | | |
| 64.5 | 2.47 | 71.1 | 4870 | .616 | .369 | | 57.5 | 3740 | .474 |
| 63.0 | 2.44 | 69.4 | 4755 | .601 | .365 | | 55.9 | 3640 | .461 |
| 54.3 | 2.06 | 59.1 | 4046 | .512 | .367 | | | | |
| 83.0 | 2.94 | 89.4 | 2825 | .697 | .270 | | 73.1 | 2970 | .542 |
| 62.9 | 2.74 | 68.6 | 2938 | .535 | .223 | | 54.4 | 2215 | .404 |
| 46.7 | 2.12 | 50.3 | 2153 | .393 | .212 | | | | |
| 122 | 3.43 | 140 | 4777 | .351 | .896 | | | | |
| 120 | 3.23 | 136 | 4647 | .341 | .920 | | | | |
| 114 | 2.84 | 127 | 4339 | .319 | .987 | | | | |
| 114 | 3.48 | 125 | 4288 | .543 | .461 | | | | |
| 104. | 3.05 | 114. | 3890 | .492 | .478 | | | | |
| 91.4 | 2.47 | 93.0 | 3356 | .425 | .508 | | | | |
| 95.8 | 3.65 | 104 | 3566 | .651 | .261 | | | | |
| 74.6 | 2.83 | 80.0 | 2746 | .500 | .248 | | | | |
| 56.5 | 1.96 | 59.6 | 2041 | .372 | .270 | | | | |

Supplemental Information to the article "Aerodynamic and Thermodynamic Characteristics of Supersonic Ramjet Engines with Mach Number between 3 and 7".

Part Ia

(1) For figures 13(a) to 13 (d), the temperature of the $C_nH_{2n}$ fuel is $T_3 0$. This figure is based on the thermodynamic properties given in Reference [1] and the combustion gas temperature is calculated for $T_1 0$ = 2000°K and M = 5 to 6.

(2) Figure 5(c) gives the $(W/a_0)_e/(W/a_0)_{T_4}$ value for equilibrium isentropic expansion of the combustion gas. Although the velocity ratio of a boundary combustion gas undergoing equilibrium expansion cannot be found from the $k_0$ formula like the variable specific heat combustion gas without a boundary, it can be seen from Figure 5(c) nonetheless that temperature, pressure and $\beta$ have rather small effects on the velocity ratio under the same $K_{0,e}$ and the same expansion pressure ratio $p_0/p_1$. The heavy curve in the figure repr·sents the velocity ratio for a given $K_{0,e}$ and under different temperature, pressure and $\beta$. For $K_{0,e} \geqslant 1.16$, the error is less than 2%.

(3) Figure 14 shows the isentropic index $K_{o,e}$ of the combustion gas. It is computed using Eq. (15) in I and the Ae value in (1) (Substitute for Figure 23 in I).

(4) Figure 15 shows the molecular weight $\mu$ of the combustion gas. Data were directly taken from Reference [1] and can be used in computing $A_{0,e}$ and W. The $\mu$ value can be read to 0.02 from the graph and this is sufficiently accurate for the above computations.

The procedures of using the above graphs are as follows:

(1) For given M and H (height) values, $p_1 o$ is obtained from Figure 1 and $T_1 0$ can be found from Figure 2.

(2) $P_3 0$ is found from $p_1 0$ and $\sigma$ values of the inlet and combustion chamber.

(3) $T_3 0$ can be obtained fairly accurately from Figure 13 for a given temperature of $T \leqslant 2000°K$

(4) Ko,e can be found from Figure 14 using $T_3 0$, $P_{3o}$, and $\beta$.

(5) $(W/a_o)_e$ is found from Figure 3(c) using Ko,e and the specified expansion pressure ratio $p_o/p_1$ of the ejector. $[(W/A_o)_{1.4}$ can be found in existing function tables].

(6) $\mu$ is obtained from Figure 15 using $\beta$, $P_{3o}$, and $T_3 0$. First, compute $A_e 2$ using the following equation

$$a_e^2 = \frac{K_{e,o} \times 9.81 \times 847.8 \times T_{3o}}{\mu}$$

and then calculate $W_4$.

All the graphs give parameters of $C_nH_{2n}$ combustion gas for the following values: $\beta$ = 0.25, 0.5, 0.8, 1, 1.5, and 2; $P_{3o}$ = (0.3), 1, 5 and 10; and $T_3 0$ = 1600 -2800°K. In case the $\beta$ ( or $P_{3o}$) value in the computation is not listed in the graph, then one can read off values for the $\beta$ ( or $P_{3o}$ ) value in the graph and plot them against $\beta$( or $P_3 0$), the quantity value is then read off for the desired $\beta$( or $P_{3o}$).

## Appendix: Calculation of the combustion gas state near the ejector throat

In ejector tube calculations, one has to find the critical parameters at the ejector throat. This calculation is a tedious iteration process for decomposing fuel gas. We now describe a very accurate method, using the existing thermal properties tables, in finding the $p^*$ and $T^*$ parameters at $M = 1$ for given total pressure $p_o$, total temperature $T_o$ and $\beta$. (Further calculations of throat area will then be possible).

First, choose three p values from the table with $p/p_o$ in the vicinity of 0.5 and carry out precise isentropic calculations respectively by three-point interpolation using $S = S_o$ and tabulated T values.

This leads to calculated T, i, w and a. Using m and the three p values, find $p_*$ and $T_*$ for $M = 1$. Obtain $i_*$, $w_*$, and $a_*$ from the table and check to see if $W_*/a_*$ is equal to 1. Generally, one cycle of such calculation leads to a discrepancy between M and 1 of less than 0.001.

Actual computations have indicated that the computation method given in Reference [21] for finding $P_*$ and $T_*$ for decomposing fuel gas really is not much more accurate than the following simple method: using given $\beta$, $T_o$ and $p_o$, find Ke,0 using the thermal properties table directly or using Figure 14 of this work, then, the following formulas allow one to compute $p_*$ and $T_*$

$$T_* = \frac{2}{K_{e,o}+1} T_o \qquad p_* = \left(\frac{2}{K_{e,o}+1}\right)^{\frac{K_{e,o}}{K_{e,o}-1}} p_o$$

$\Delta T_*/T_*$, $\Delta p_*/p_*$ are generally within 2%. Further calculation on throat area $A_*$ may have a greater error (5% or more) because of the accumulation of errors.